D. Transitional Interference Statistics

The commission has requested statistics of the interference created to a DMSP ground station from the time a DMSP satellite would change frequency band segment to the time all the Little LEO satellites could be re-commanded. This is a multidimensional problem, complicated by the potential world wide DMSP user locations and Little LEO command sites. In order to address these statistics, we have first assumed that a DMSP satellites is re-commanded while in view of Sunnyvale, CA. We have then considered two cases, the command station "Fence" sites consisting of San Diego, Miami, Sinnamary (French Guinea), and Santiago (Chile) and the "90 minute" site locations defined by Seattle, Miami, Valdiva, Tokyo and Melbourne. We have also assumed that the initial frequency assignments made to Leo One USA satellites were such that they would not interfere with the DMSP satellite. This implies that after the DMSP frequency change, every Leo One USA satellite that comes in contact with the DMSP satellite will cause interference since it will now be on the wrong frequency.

Since the frequency change will be completed within an orbital period, over much of this time, the Leo One USA satellites will be interfering with the DMSP satellite. It should also be noted that this interference, in general, does not extend over the entire DMSP coverage footprint. Thus, many users would still receive DMSP downlinks interference free. Further, for any individual DMSP user station, this interference would only occur once. Averaged statistics of interference to a single user are therefore difficult to interpret if not somewhat meaningless.

Figure 16 shows the DMSP and Leo One USA satellite ground traces for one Leo One USA orbital period (104 minutes) that were used for this evaluation. Figure 17 shows the Leo One USA satellites in contact with this DMSP satellite over an orbital rev. The satellites are numbered sequentially from 1 to 48 starting from the first satellite in plane one. The contact time is computed for a 5 degree DMSP coverage to a horizon coverage Leo One USA footprint. We believe this is a reasonable worse case situation in that the DMSP satellite is in a retrograde orbit moving north and west away from the Leo One USA satellites that are re-commanded at the start of this simulation. It takes approximately half a rev before the DMSP satellite contacts the recently commanded Leo One USA satellites which are moving eastward. Since Leo One USA's horizon coverage is 100 percent between approximately ±70° latitude, the DMSP satellite can only operate interference free to high latitude (polar) ground stations until the satellites are re-commanded.

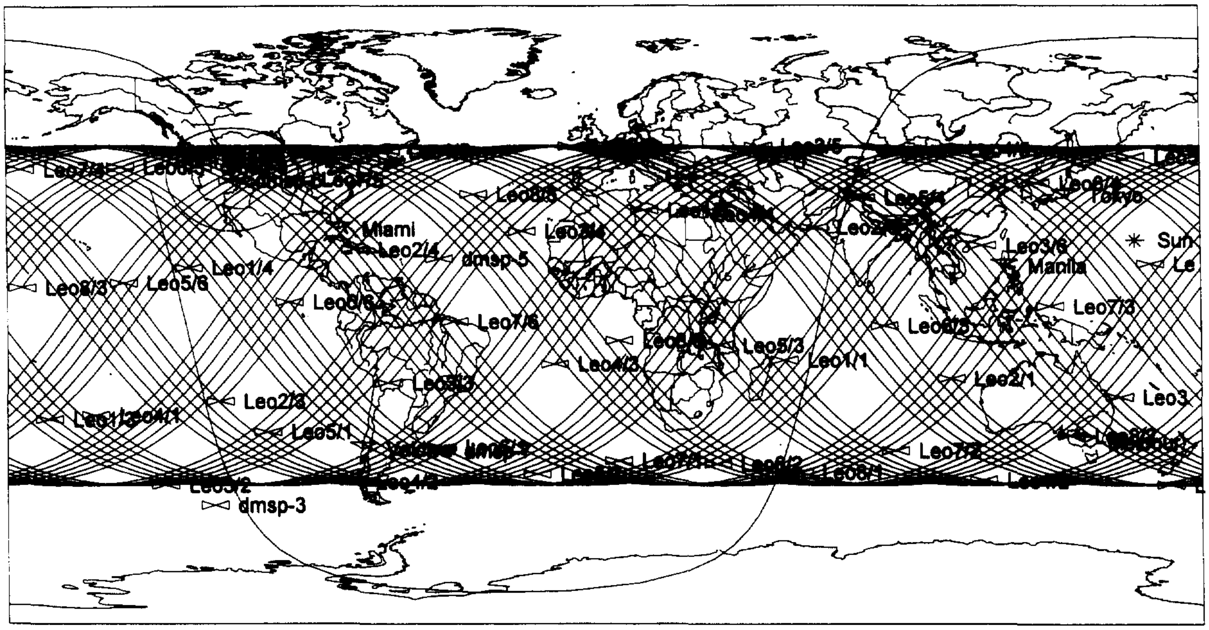

Figure 16. DMSP and Leo One USA Ground Traces.

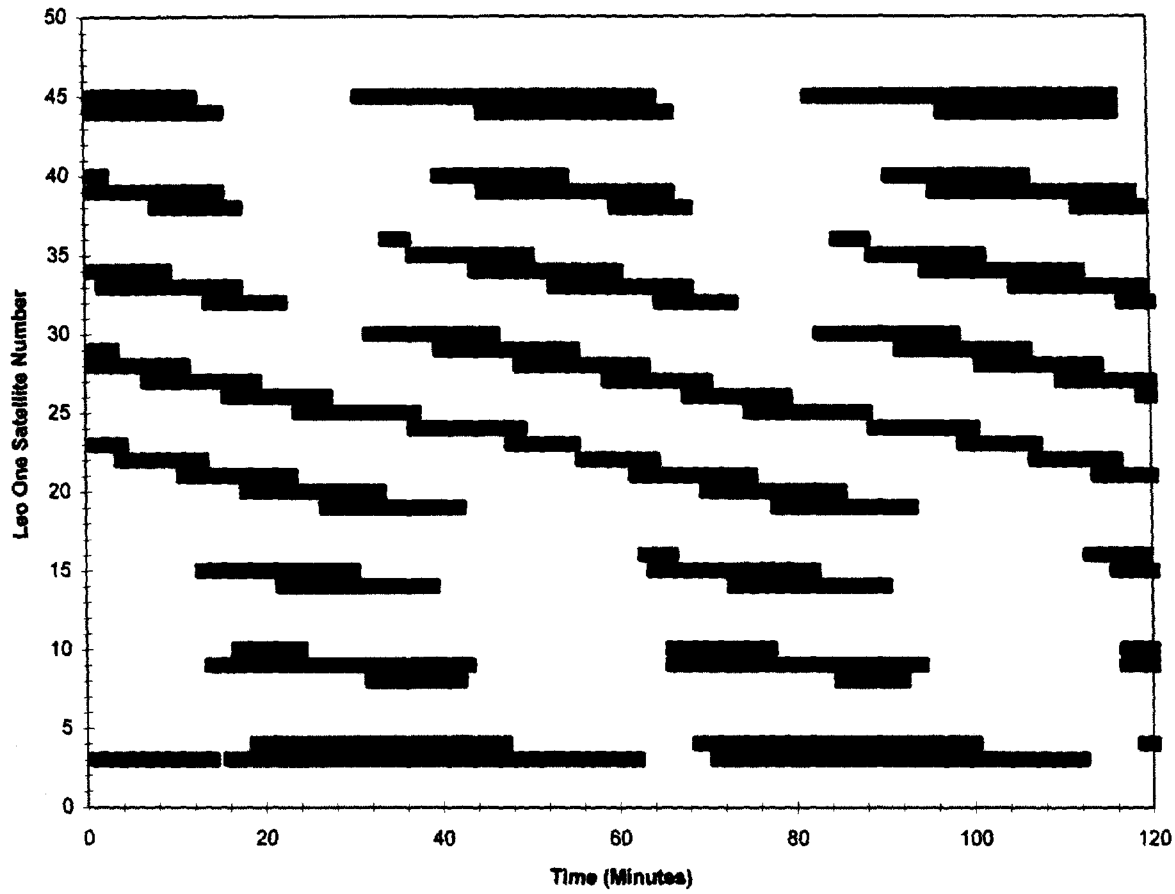

Figure 17. Leo One USA satellites In Contact With DMSP Satellite.

1. "Fence" Site Results

Figure 18 shows the Leo One USA satellite command times for a minimum 10° elevation angle. This figure starts from the DMSP Sunnyvale contact time and assumes commanding of the Leo One USA satellites begins immediately as they contact the command stations along the "fence."

Figure 19 shows the timeline for footprint overlap conflicts as a function of time. As indicated, after approximately 55 minutes half of the satellites have been commanded to their new frequency assignments. Only 8 satellites cause interference after 55 minutes. The last satellite conflict ceases at 94 minutes. After 104 minutes all satellites have been commanded to new frequencies and all possible conflicts cease.

Figure 20 through Figure 25 shows the extent of the DMSP coverage footprint overlap each 20 minutes over this one rev period. The DMSP coverage is shown as 5° and the Leo One USA coverage is shown as 0°. As indicated, the loss in coverage area for the DMSP satellite shrinks dramatically after approximately 55 minutes, or approximately one-half the Leo One USA orbital period. However for the worse case situation, the sun synchronous retrograde orbit maintain a conflict as a result of its polar and westward motion. If the satellite had been commanded on its downward leg approximately 12 hours later, this situation would have been less severe.

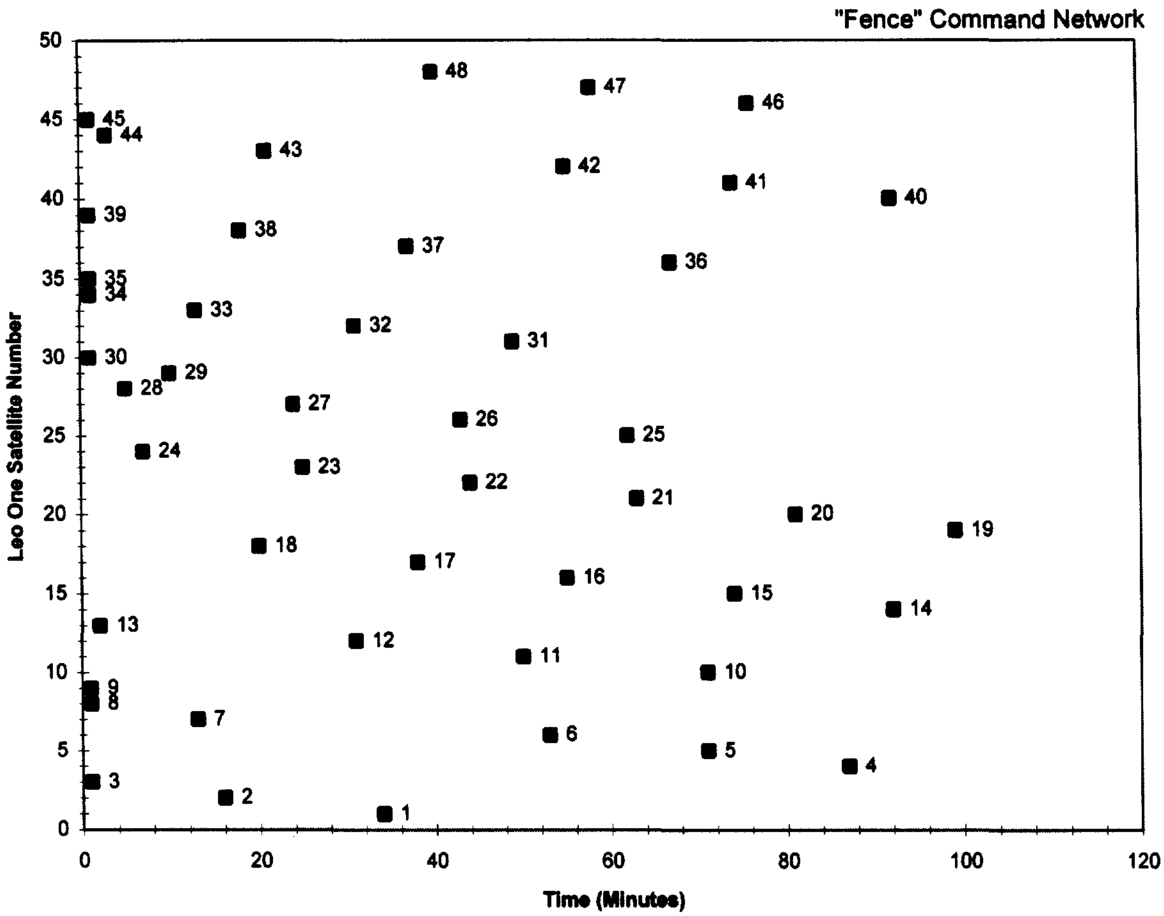

Figure 18. Leo One USA Satellite Command Times Over a 104 Minute Period With "Fence" Command Sites.

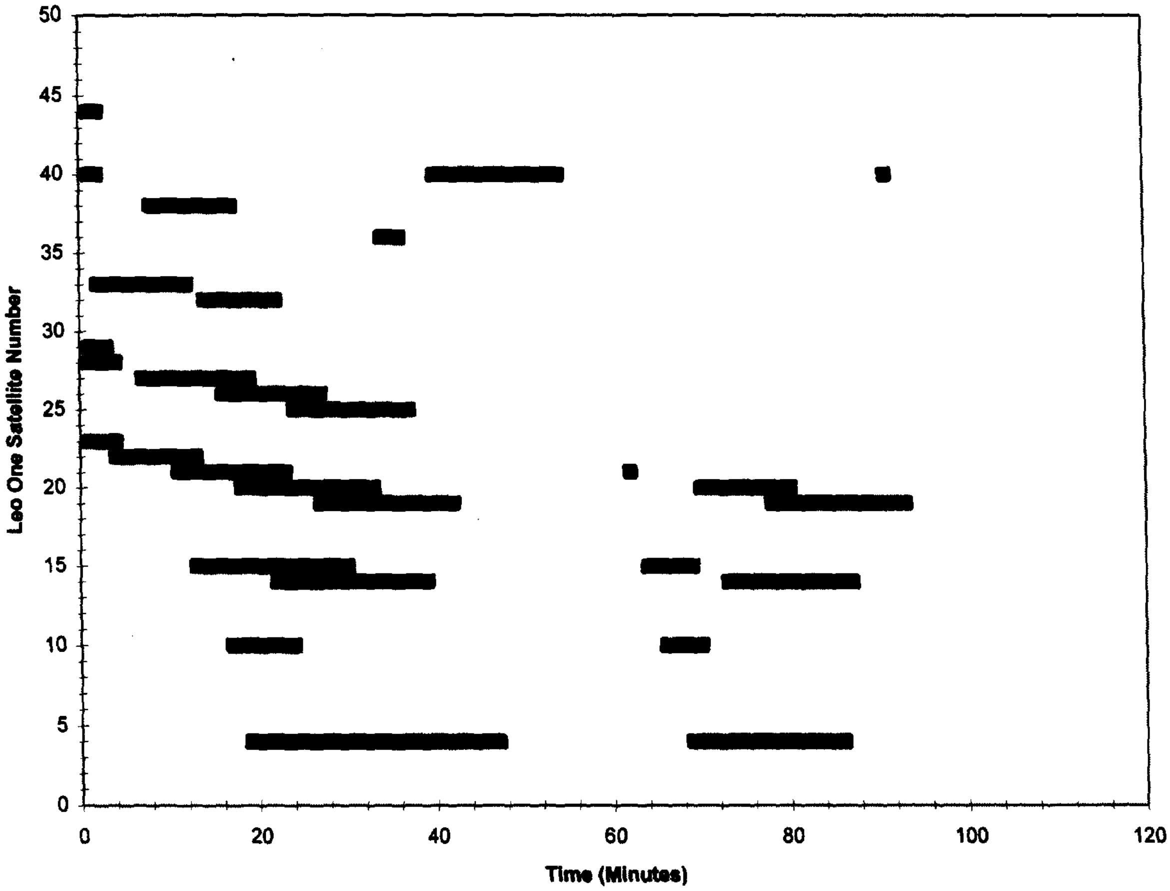

Figure 19. Transitional Interference Times To DMSP Using "Fence" Command Sites.

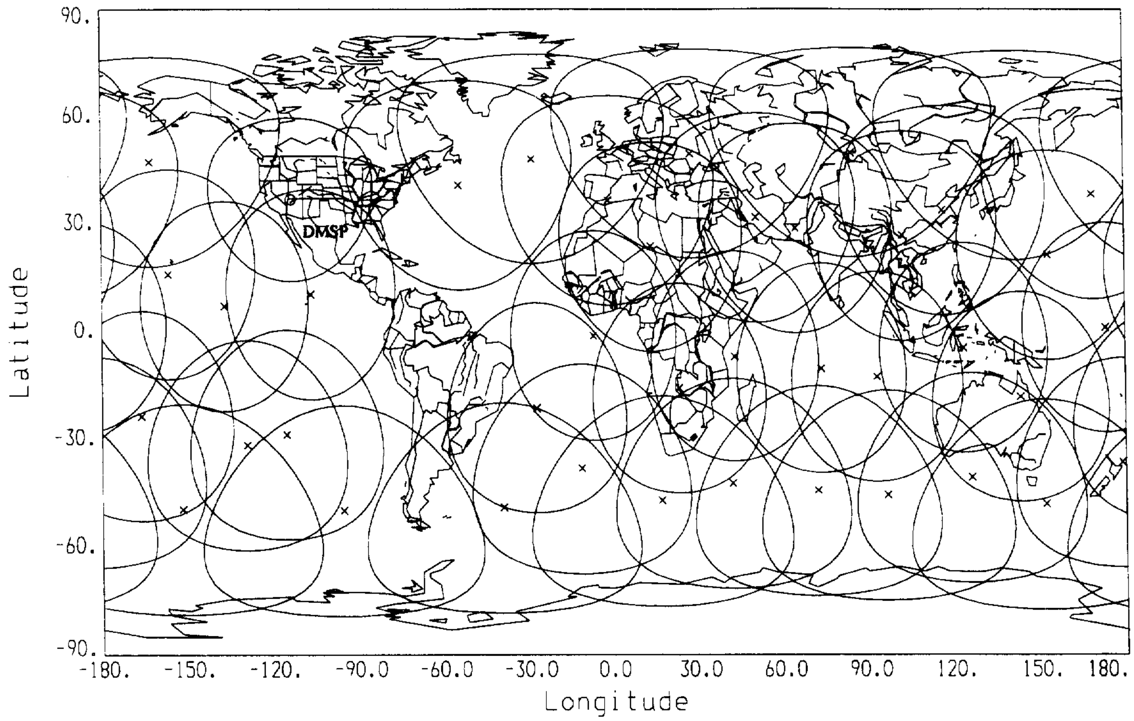

Figure 20. Leo One USA Interference Footprint Overlaps With DMSP At 0+ Minutes.

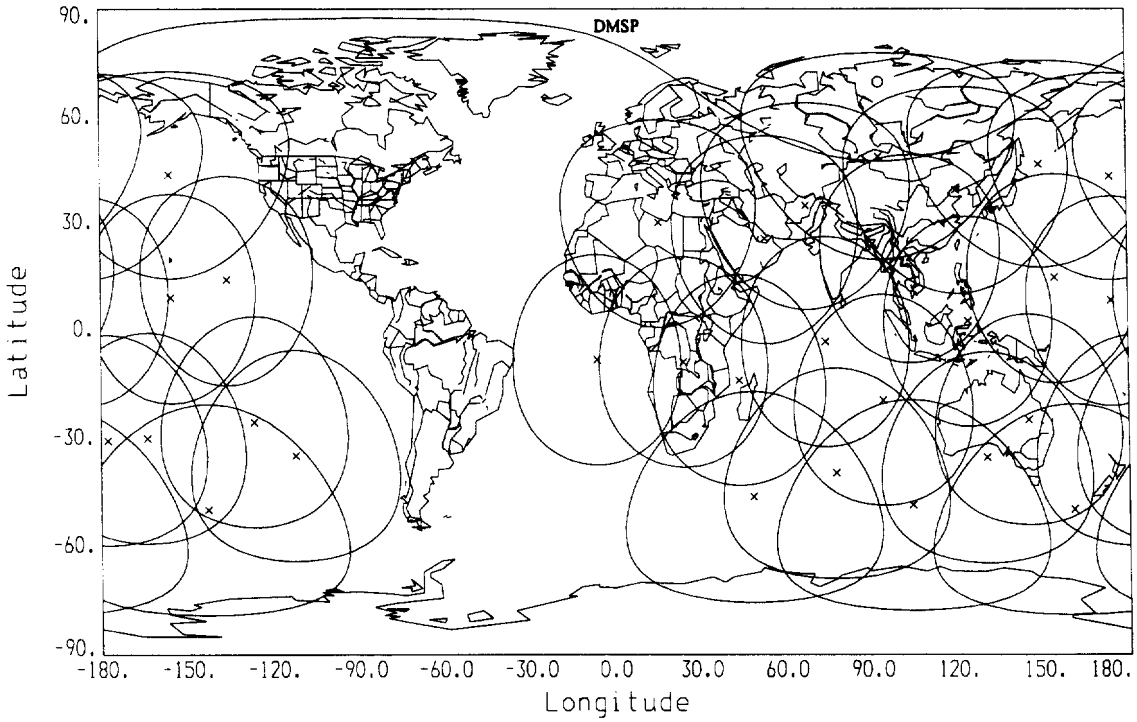

Figure 21. Leo One USA Interference Footprint Overlaps With DMSP At 20 Minutes.

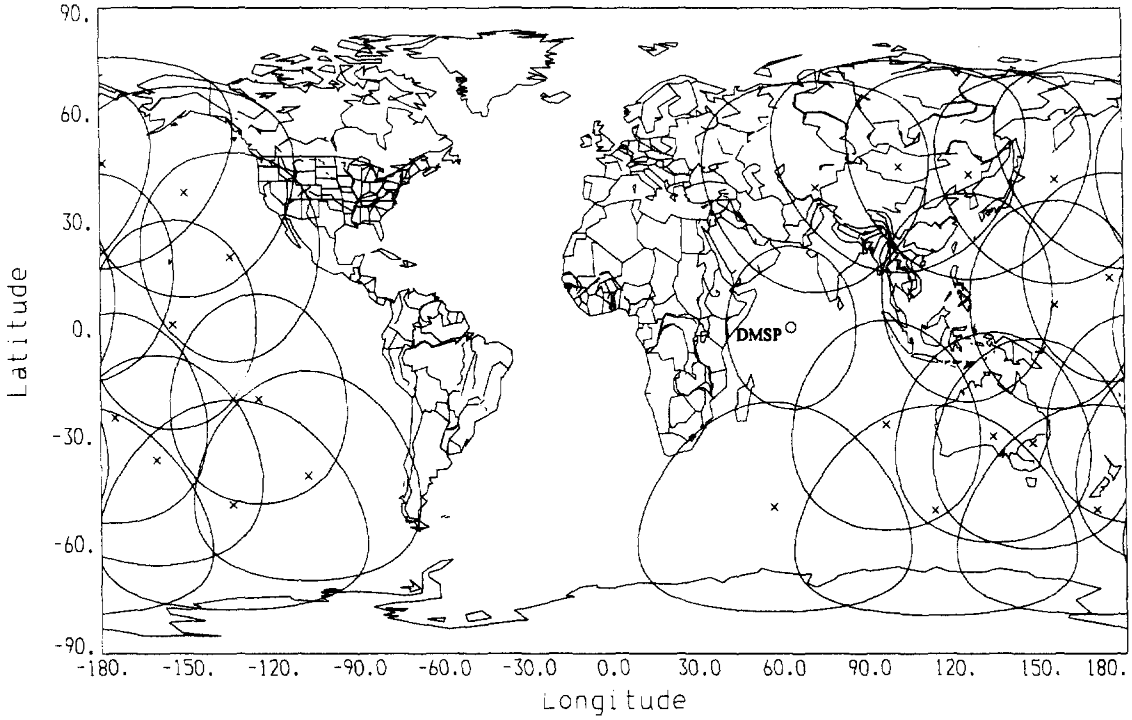

Figure 22. Leo One USA Interference Footprint Overlaps With DMSP At 40 Minutes.

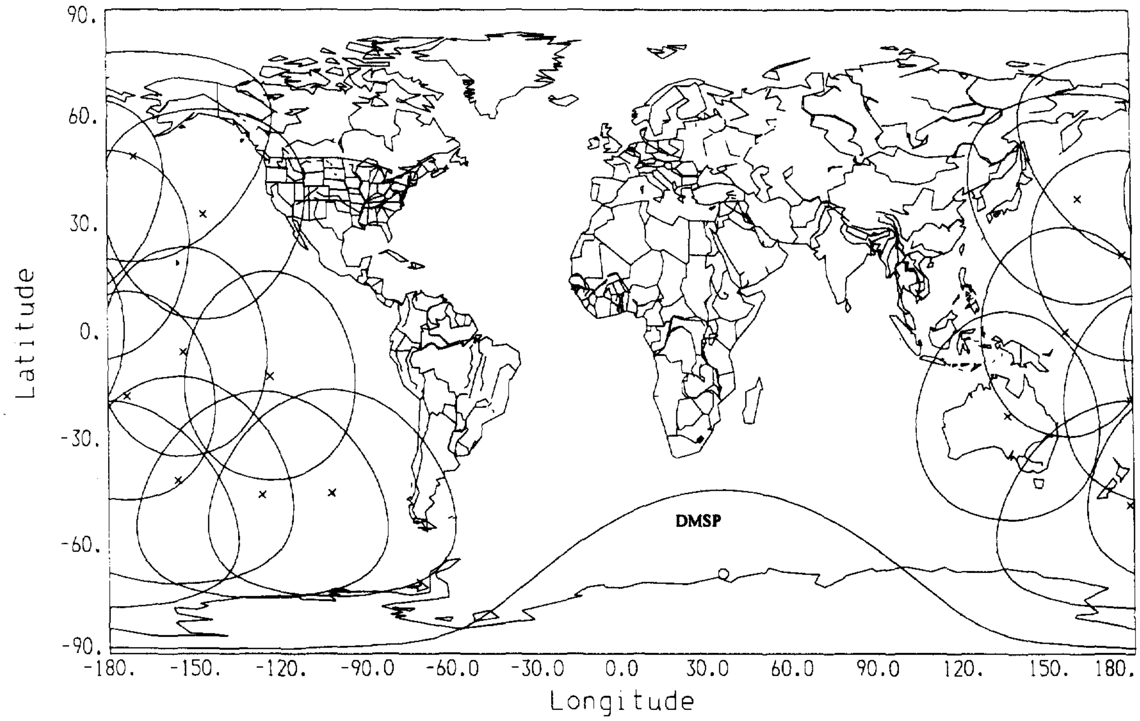

Figure 23. Leo One USA Interference Footprint Overlaps With DMSP At 60 Minutes.

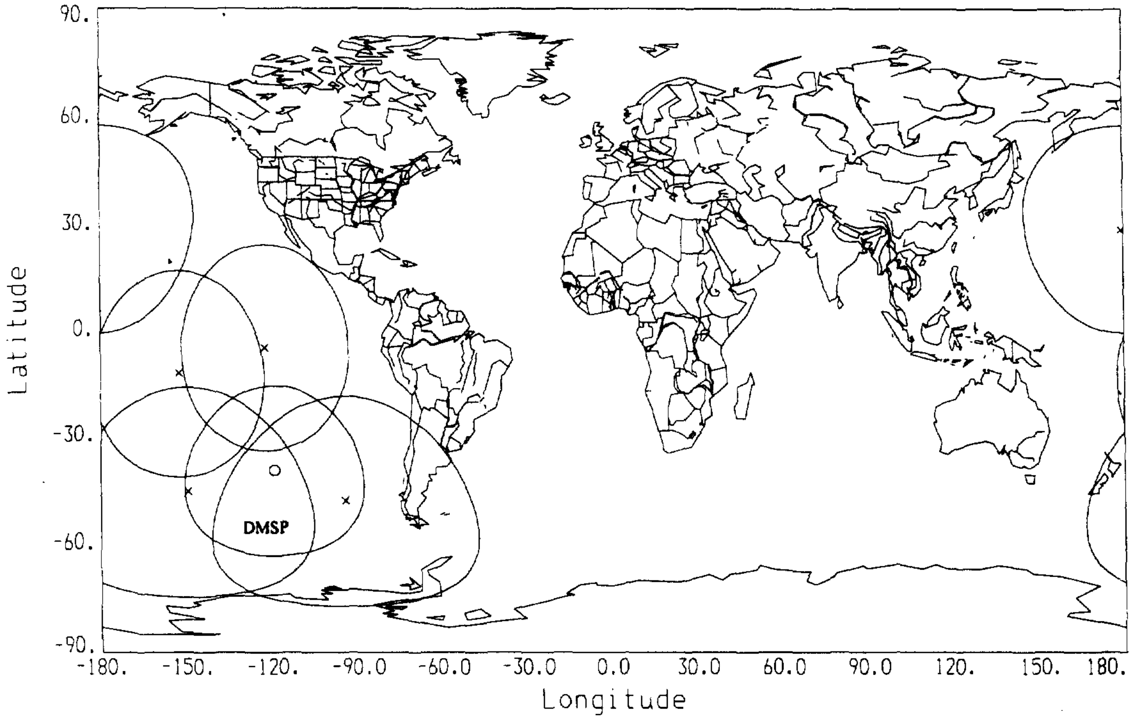

Figure 24. Leo One USA Interference Footprint Overlaps With DMSP At 80 Minutes.

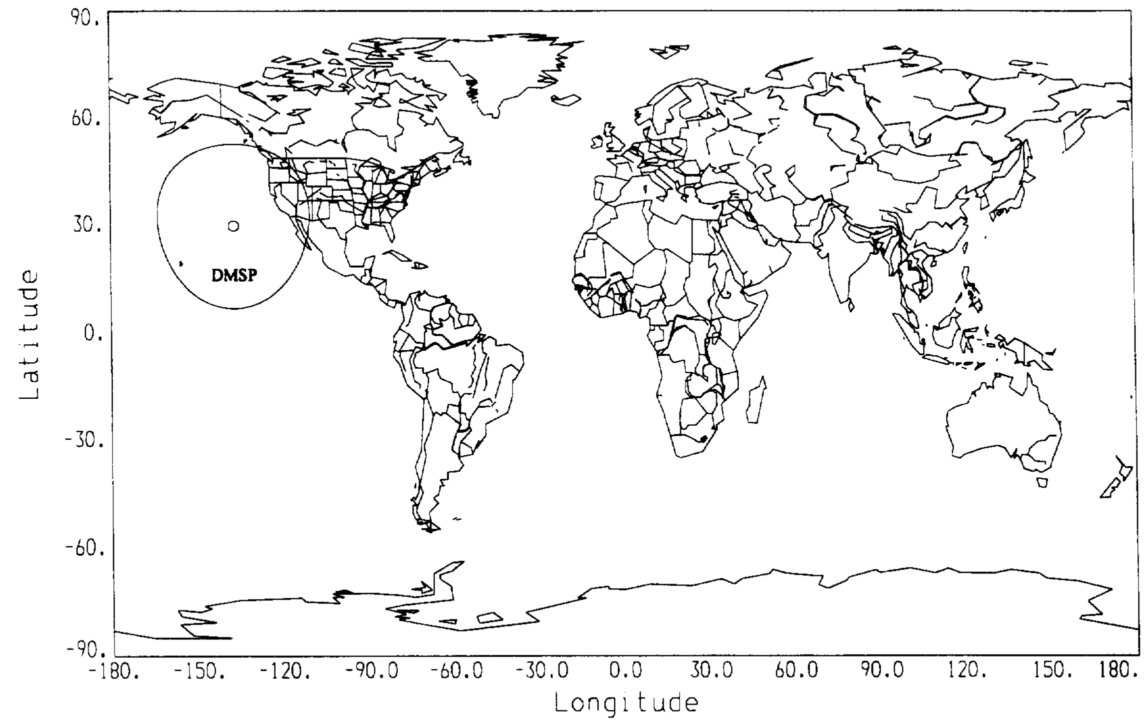

Figure 25. Leo One USA Interference Footprint Overlaps With DMSP At 100 Minutes.

2. <u>"90 Minute" Site Results</u>

Figure 26 shows the Leo One USA satellite command times for the "90 Minute" command sites using a 10° minimum elevation angle. This figure starts from the DMSP Sunnyvale contact time and assumes commanding of the Leo One USA satellites begins immediately as they contact the command stations. This result is for the five 90 minute command site locations. As shown, for this particular starting time, all satellites are commanded within 60 minutes. Again, the potential interference times are given by Figure 27.

Figure 27 shows the timeline for footprint overlap conflicts as a function of time. As indicated, after approximately 53 minutes all of the conflicting Leo One USA satellites have been commanded to their new frequency assignments. After 40 minutes the area impacted is small.

Figure 28 shows the extent of the DMSP coverage footprint overlap at 20 minutes and Figure 29 shows the coverage overlap at 40 minutes. As indicated, the loss in coverage area for the DMSP satellite shrinks dramatically after approximately 40 minutes and is grouped in one longitude zone. Again, all satellites have not been commanded until 60 minutes have elapsed in this example.

These limited simulation examples are intended to indicate the nature of the transitional interference. Many variations are possible. However, the results presented here are believed to be representative. Again, it would seem that the one orbital period update time would be adequate for most all purposes. More importantly, all interference can be avoided through pre-planned and coordinated frequency changes.

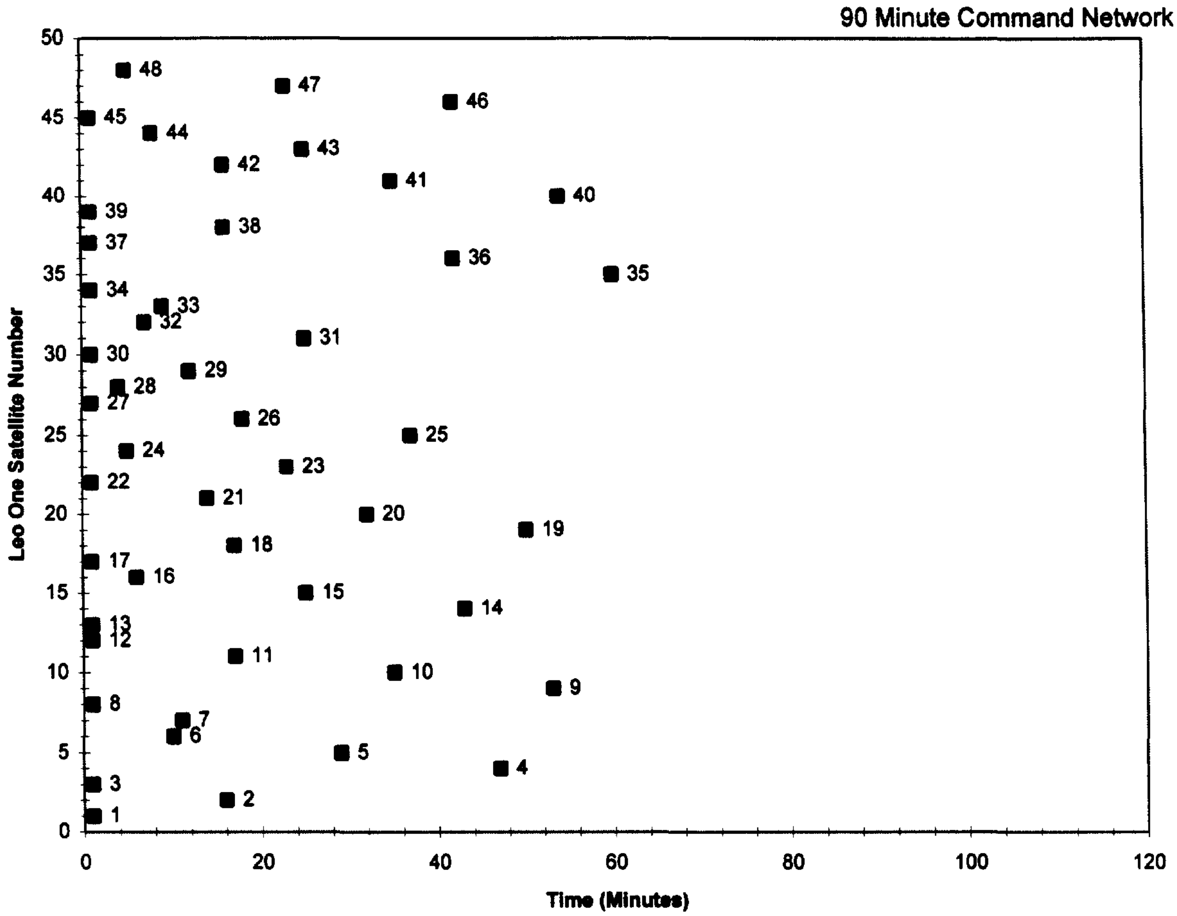

Figure 26. Leo One USA Satellite Command Times With "90 Minute" Command Sites.

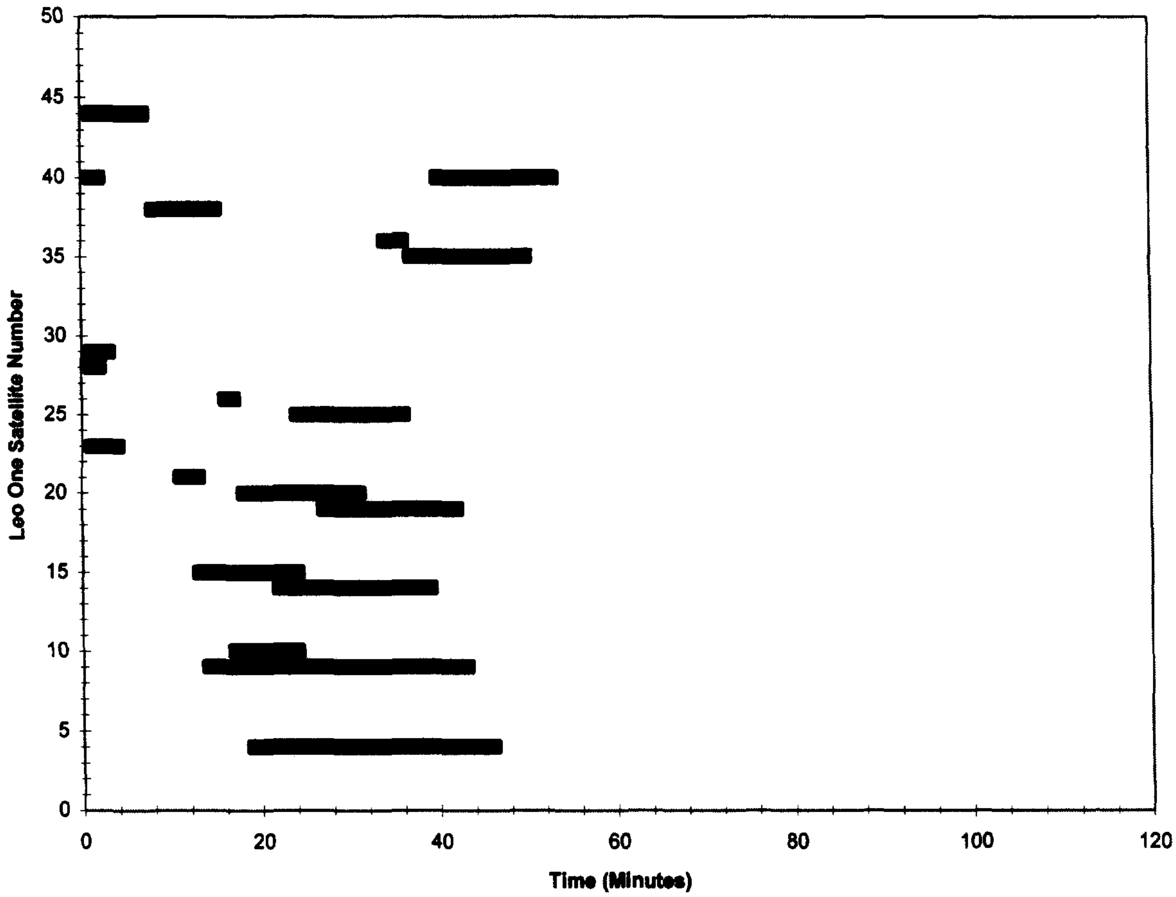

Figure 27. Transitional Interference Times To DMSP Using "90 Minute" Command Sites.

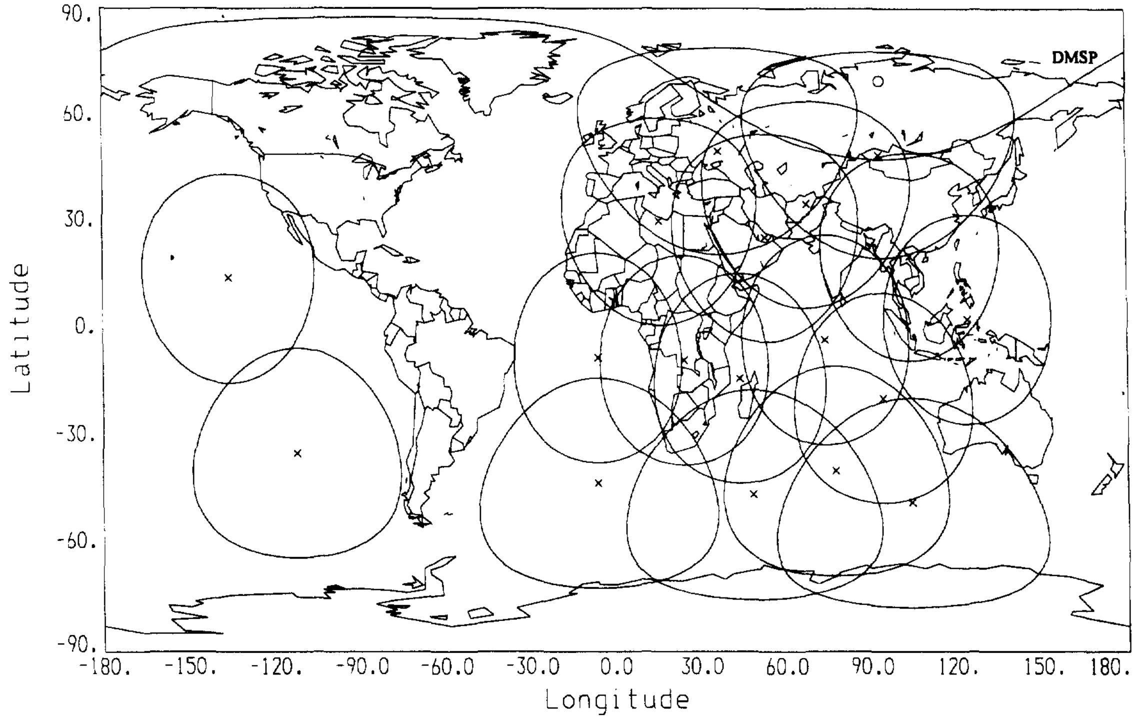

Figure 28. Leo One USA Interference Footprint Overlaps With DMSP At 20 Minutes - "90 Minute" Command Sites.

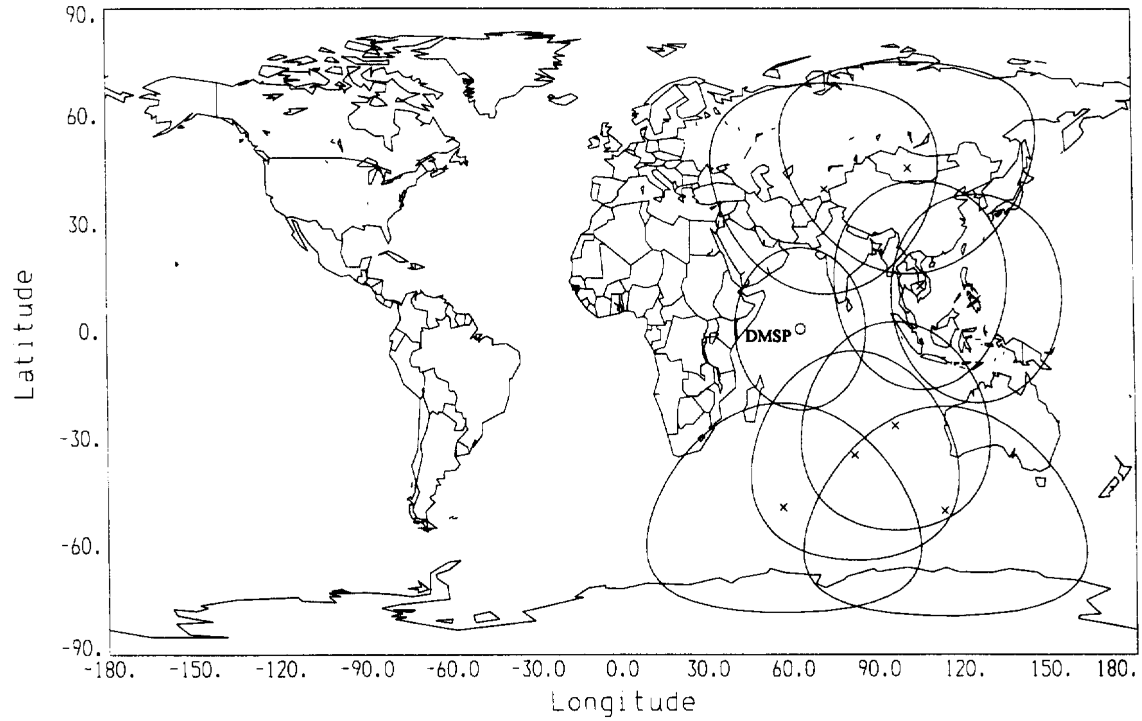

Figure 29. Leo One USA Interference Footprint Overlaps With DMSP At 40 Minutes - "90 Minute" Command Sites.

E. <u>Accurate Ephemeris Prediction</u>

There are two parts to the prediction of potential coverage overlaps of DMSP satellites with a Little LEO constellation coverage footprints. The <u>first part</u> deals with the determination of the satellite orbit element sets (or ephemeris) and the second part deals with the forward propagation of this element set to future locations of these satellites. This prediction accuracy determines the need to grow the exclusion radius to guarantee no interference to DMSP. Here we assume the defined exclusion radius has been defined in terms of the necessary minimum elevation angle contours from the DMSP and Little

LEO satellites. The prediction uncertainty then effectively adds to the required DMSP exclusion radius. For instance, 10 km in coverage location uncertainty would increase a 5 degree elevation angle coverage exclusion zone from 2612.9 kilometer radius to 2622.9 kilometers on the surface of the earth. This is equivalent to a 4.89 degrees elevation coverage exclusion zone for DMSP instead of 5.0 degrees. This difference is not significant in terms of the availability impact to Little LEO coverage.

The orbit determination process accuracy is a function of the observation interval and the accuracy of the observables (range and elevation angle for NORAD radar tracking) and the sparseness of the observations. Typically, commercial satellites and most DoD satellites provide their own orbit determination through ranging from surveyed ground sites. Newer and potentially more accurate near real time approaches have evolved through the use of GPS. Point positioning with GPS is as accurate in low orbit as on the ground, typically 50 to 100 meters[5] for single frequency C/A code users with nominal levels of GPS selective availability (a degradation imposed by DoD on GPS civil user accuracy). Corresponding velocity estimates may approach 0.5 m/s accuracy. The accuracy achieved is a function of the receiver design and care taken in time tagging data.

These instantaneous measurements, however, may be inadequate for orbit prediction purposes. Classical dynamic orbit determination exploits orbital mechanics and filtering theory to yield a stable and accurate orbit solution from generally sparse and noisy measurements. This approach is required for conventional tracking systems. In dynamic orbit determination, the orbit model is derived from models of the forces acting

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[5] T. Yunck, "Orbit Determination", Global Positioning System: Theory and Applications. Vol. II, AIAA Vol. 164, 1996, pp 559-589.

on the satellite and the laws of motion. Highly accurate models include the satellite physical properties. The major forces include gravity, aerodynamic drag and lift, solar radiation pressure, and active thrusting (once or twice per orbit). Lesser contributions may come from outgassing, satellite thermal radiation, sunlight reflected from the Earth, and electromagnetic effects. The force and satellite models are used to compute a model of satellite acceleration over time, from which by double integration, a nominal trajectory is formed. In principle, all that is required to produce the orbit solution is to determine the two vector constants of integration, position and velocity, at one time point. This is done through an estimation procedure that finds the best estimate of this epoch state (usually through minimizing the mean square fitting error) for which the resulting model trajectory best fits the tracking data according to some optimality criterion.

NASA has developed what has been termed kinematic solution approaches that enable much more precise orbit determination using GPS measurements. For instance, predictions with the space shuttle has shown the 3-D position error of 43 meters using dynamic solutions, 28 cm with single frequency kinematic solutions and 3 cm with dual frequency kinematic solutions after 8 hours of arc measurements[6]. The achievable accuracy using direct real-time GPS approach is shown in the figure below (Figure) which is a function of orbit altitude. Actual performance will depend on specifics of the GPS tracking configuration and satellite dynamics, but the figure does indicate that

[6] T. Yunck, "Orbit Determination", Global Positioning System: Theory and Applications. Vol. II, AIAA Vol. 164, 1996, p 588.

satellite location techniques are available that are more than adequate for interference avoidance requirements.

We would recommend the commission not dictate a required method, as virtually all reasonable approaches should be more than adequate for interference avoidance. Depending upon the method chosen and the resulting accuracy, the coverage error radius must be adjusted accordingly. This can be coordinated with NOAA to mutual satisfaction.

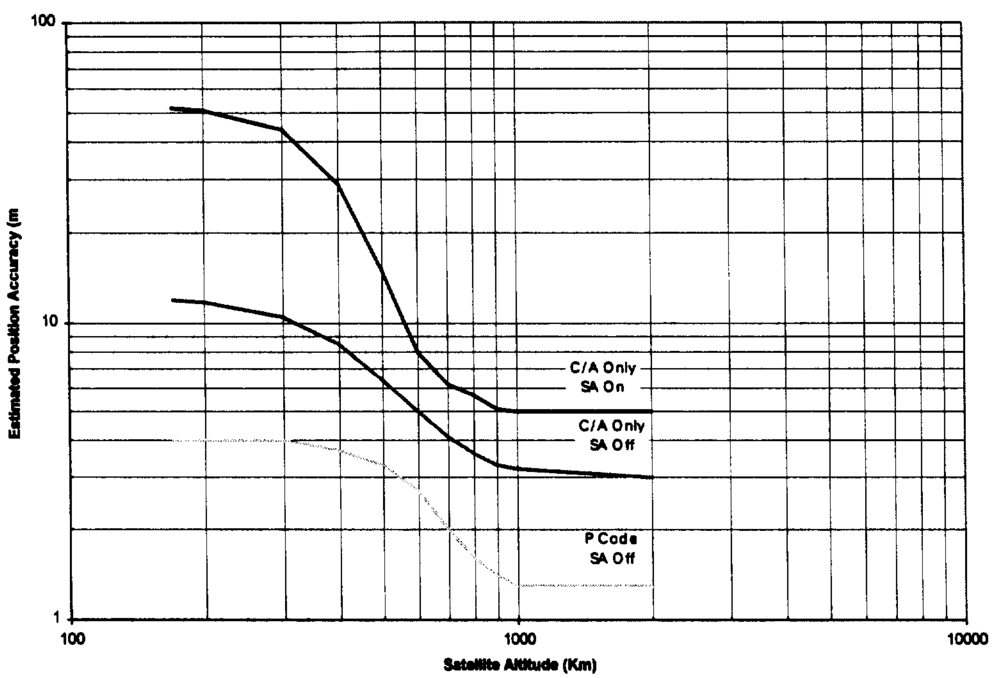

Figure 30. Estimated Orbit Accuracy's Achievable Using Real-Time Direct GPS Techniques[7].

[7] Adapted from T. Yunck, "Orbit Determination", Global Positioning System: Theory and Applications. Vol. II, AIAA Vol. 164, 1996, p 588.

The second part of the problem is the forward time projection of position location assuming no further real-time data is available following an element set estimation. This is accomplished through the use of orbit propagators. Orbit propagation can be accomplished through analytical or numerical methods. The more advanced and accurate methods will use numerical integration to generate the ephemeris. Numerical methods include the perturbation effects from the earth's oblateness, atmospheric drag (drag as a function of altitude and temperature), sun-lunar perturbations, solar radiation effects, and other large body effects (i.e., Jupiter, moon, etc.) Generally, the Cowell method is the choice of numerical methods to be used by propagators. The following are some propagators using numerical methods that are commercially available.

- *Orbital Workbench*: This is a fairly accurate tool that uses the Cowell method. It has been used on DoD projects to verify NORAD data within 100 meters. This tool is over 5 years old and the newer version is even more accurate with updated atmospheric drag tables. However, the latest version is still in a beta form and not ready for release.

- *WS/PODS*: This tool is another Cowell numerical integrator that has accuracies within 1.0 meter of the true orbit with a fitting accuracy of 10 cm. PODS is a very reliable tool and takes into consideration all of the gravitational forces. It is an active tool that allows and performs updates to the drag tables. The tool is flexible enough that the integration step size and force models can be changed and updated. LEO-One has this tool currently running under STK, but it can also run as a stand alone batch file. The difference is that the batch file does not have the pretty graphics that STK has. The stand alone is approximately $10,000 and the STK is $20,000. GPS-MET is using this tool currently.

- *HPOP:* This propagator uses a 7th order Runge-Kutta method to propagate its orbit. It is accurate to within 10 meters of the original orbit. This tool is run under STK and costs approximately $20,000.

Analytic methods are also used to perform orbit propagation. These methods are not as accurate as numerical and usually do not include perturbation effects except the J2 effect. The following are analytic propagators:

- *Astrovis*: The best propagator that this tool has is determining the Keplerian orbit with the J2 effect included. Not realistic for use with any kind of accuracy.

- *PCSOAP*: This tool has different propagators. It uses the SGP, SGP4 and SDP4 NORAD propagators that are old and not very accurate. It also uses the Keplerian propagator like Astrovis but states a disclaimer that they will not guarantee its accuracy past 24 hours. They also mention a Low Thrust propagator that includes most perturbation effects. Not all that advanced from Astrovis.

If the accuracies that NOAA/DMSP/DoD requires of Leo One USA are very tight then it is reasonable to choose a tool that uses a numerical integration method to propagate the orbit. The costs can be high, however. All the propagators are set up to read two-line element sets as well as state vectors and propagate accordingly.

It is our understanding that NOAA and DMSP currently rely on NORAD radar tracking for orbit determination and SGP4 propagators using mean two line element set for prediction. We would recommend that more precise orbit determination and prediction means be used in the future. NORAD's prediction techniques are typically not very accurate, especially when using a mean two line element set. NORAD's requirement for NOAA is typically set to predict within an accuracy of 5 km with 90 percent confidence. As a result of this requirement, the period over which the predicted orbit is accurate is typically two weeks. The accuracy of orbit propagation is dependent upon the accuracy of the orbital elements provided by the tracking system (observables), the precision to correct the orbital emements for errors (orbit determination), and the precision of predicting the spacecraft's orbit (orbit propagation). This is discussed further below.

Observables

NORAD tracks satellites from various ground stations that obtain azimuth, elevation, range, and range rate data. The observable's require multiple passes to determine its orbit within an accuracy of 5 km. The observables obtained from each of the ground stations are different due to location, environment, and equipment. The errors incurred from a station are weighted and factored into the orbit calculations for the different satellite observables. As a result, these errors are propagated with the orbit and affect the length of period for which a predicted orbit is good.

Satellite ranging provides an alternative method for generating observable data. This approach has been performed so long that a reliable system can achieve accuracies less than 1 meter.

Orbit Determination

NORAD uses the satellite observables to generate a Two Line Mean Element Set (2LMES). A mean element set is defined as the motion of the orbit over a span of time. When taking data over a period of time and averaging the data, the element set is no longer defining the true orbit, but some integral of it. Therefore, the orbit that is propagated is not the true orbit.

For more precise orbits an osculating element set is used. Osculating elements describe a true Keplerian orbit instantaneously tangent to the motion of the true orbit. For instance, GPS tracking data can provide a State Vector for the orbit determination method. The state vector is the position and velocity of the spacecraft at an instant in